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CALCULATION OF AERODYNAMIC HEAT TRANSFER FOR BLUNT-NOSED THIN WINGS AT DIFFERENT ANGLES OF ATTACK AT SUPERSONIC SPEEDS

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A method for calculating aerodynamic heat transfer for thin wings at angles of attack is provided, based on the assumption of small cross flow and the calculation of an infinitely long cylinder as described in this paper. The method is valid in the supersonic and low hypersonic speed range. The method does not require calculating streamline details and can be used for various sweptback wings at moderate angles of attack or yaw angles. The theoretical calculation is compared with the experimental result.

Symbols

C_p -- Specific heat at constant pressure, kilocalories/kilogram force·degree;

M -- Mach number;

p -- Pressure, kilogram force/square meter;

q -- Heat flow rate in kilocalories/square meter·second;

s -- The object surface line as measured from the stationary-point line, parallel to xy plane, meters;

$\bar{s}$ -- The object surface line perpendicular to the leading edge as measured from the tip of the wing leading edge;

θ_1 -- The included angle of line s and along the direction of the generatrix of the leading edge, degrees;

μ -- Coefficient of viscosity in kilogram force·second/square meter;

A -- Sweptback angle of the wing leading edge, degrees;

ρ -- Density in kilogram force·second^2/meter^4;

ϕ -- Circular angle at leading edge of cylinder as measured from tip of the wing leading edge, degrees;

T -- Temperature, K;

V_∞ -- Velocity of incoming flow, m/s;

u_c -- Velocity at the outer fringe of the boundary layer, m/s;

* Numbers in the margin indicate pagination in the foreign text.

-- The included angle along the direction of the leading edge generatrix of streamline ℓ, degrees.

Footscripts

0 -- Value of stationary point; ∞ -- Value of incoming flow;
e -- Value of outer leading edge at the boundary layer; s -- Value of the stationary-point line;
w -- Value at the wall surface.

I. Foreword

Calculation of aerodynamic heat transfer for thin wings at various angles of attack is a problem concerning present-day flight vehicle designers. It is an effective method shown in practice, relying on the assumption of small cross flow in calculating heat flow for an object at an angle of attack. For somewhat longer than the past decade, this calculation technique became relative well-suited to design needs thanks to several studies by F. R. Dejarnette et al. [1, 2]. However, in all these methods, calculations begin from the front stationary point; these methods are adaptable for calculating the heat flow of a blunt body, not for direct application to thin wings with a leading edge that has a very small radius. Even earlier, there have been "two-dimensional" formulas [4] for calculating heat flow perpendicular to the leading edge of plane wing at various angles of attack. Although only laminar flow can be calculated, the calculation is very simple.

Starting from an assumption of small cross flow, the paper /399
derives a working formula for the heat transfer rate of an infinitely long cylinder. On this basis, one version of the method for calculating heat transfer rate of thin wings is proposed. The advantage of this working formula for an infinitely long cylinder is the following: one is required only to calculate the direction of inviscid streamlines, and is not required to calculate concretely the stream-

line shape. Thus, not only are the calculation procedures much simplified, but also the difficulty of not easily to control the streamline distribution at large angles of attack and large yaw angle δ is overcome. To adapt this method version to sweptback wings with different configurations, this paper presents the related geometric relations derived from vector analysis.

II. General Presentation of Calculation Method

(1) Analysis: By adopting the assumption of small cross flow and neglecting the influence of secondary flow, in the streamline coordinate system the three-dimensional boundary layer equations can be transformed into axially symmetric flow [1]. Hence, by using the axial symmetry analogy, the axial symmetry flowing heat transfer formula can be used to calculate heat flow around an object at various angles of attack. In other words, the calculation is carried out along an inviscid streamline ℓ, and the lateral direction scale factor h_2 replaces the diametral radius r. The numerical result concerning the drift infinitely long cylinder in reference [3] can demonstrate the feasibility of the small cross flow assumption.

To an infinitely long cylinder with equal-cross-section (Fig. 1) with drift, shapes of all inviscid streamlines on the surface of an object are the same; between the adjacent streamlines, there is only the displacement $\Delta\bar{z}$ along the $\bar{z}$ direction; $\Delta\bar{z}$ is an constant. If Δn stands for the vertical clearance between two streamlines, by definition, h_2 is approximately equal to the ratio between Δn and $\Delta\bar{z}$ [3].

$$h_2 \approx \frac{\Delta n}{\Delta \bar{z}} = \frac{\Delta \bar{z} \sin\theta}{\Delta \bar{z}} = \sin\theta$$

Therefore, the heat transfer rate formula (laminar flow) is as follows:

$$\left(\frac{q}{q_e}\right)_1 \sim \frac{\rho_e \mu_e u_e \sin\theta}{\left[\int \rho_e \mu_e u_e \sin^2\theta d\ell\right]^{1/2}} \qquad (1)$$

In practical problem, one often wishes to calculate heat flow of line s on the surface of an object. According to the relation between the corresponding line s and ℓ, obviously, $\sin\theta_1\, ds = \sin\theta\, d\ell$.

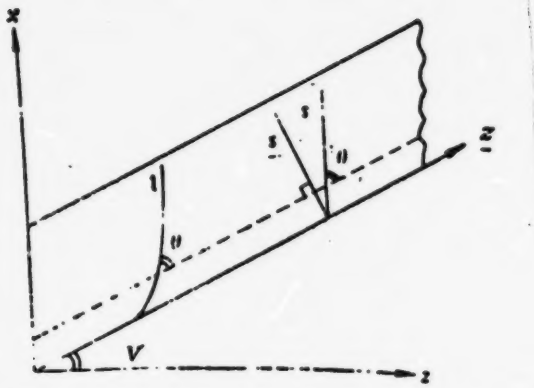

Fig. 1.

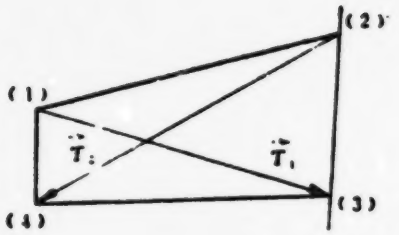

Fig. 2.

Then Eq. (1) is transformed to become the following

$$\left(\frac{q}{q_0}\right)_1 \sim \frac{\rho_e \mu_e u_e \sin\theta}{\left[\int \rho_e \mu_e u_e \sin\theta \sin\theta_1 \, ds\right]^{1/2}} \qquad (2)$$

The calculation is thus simplified: the shape of streamline ℓ need not be calculated; one needs only to calculate the corresponding θ and θ_1 along s, the equivalent result can be obtained as Eq. (1) by using Eq. (2). Likewise, a similar relation obtains in the problem of turbulent flow. /400

As mentioned above, the working formula along line s can be approximately extended to blunt-nosed thin wings. As shown by analysis, along the thin wing, the calculation is conducted based on the difference along s (from along ℓ) mainly depending on different pressures of their corresponding points. However, this kind of pressure nonuniformity will tend to moderate with higher Mach number [5]. Moreover, when Eqs. (1) and (2) are used to calculate heat flow at the same point, if it is assumed that h_2 relies only on local conditions, the external fringe value of the boundary layer is determined by the local pressure; then the difference between the equations lies only

in the integration of the denominator. Here, the square root of the integration is used; in turbulent flow, it is the fifth root. Hence, one can predict that in general applying the relations for an infinitely long cylinder will not lead to larger errors.

(2) Shape of thin wings and flight attitude: A thin wing is composed of two parts: the wing's leading edge and the wing planform one assumes that the leading edge is cylindrically shaped; the wing planform is approximated with several plane elements. The thin wing shape is described by the radius R_s of the leading edge, the sweptback angle A, and the angular point coordinates of various plane elements of the wing planform.

In the thin wing coordinate system xyz, the component of unit normal vector $\vec{n}$ on the leading edge of the wing is

$$n_x = -\cos\varphi\cos\Lambda,\ n_y = \sin\varphi,\ n_z = \cos\varphi\sin\Lambda$$

The normal vector [9] on the wing plane is $\vec{N} = \vec{T}_2 \times \vec{T}_1$ (Fig. 2); the unit normal vector $\vec{n}$ is marked as $\vec{n}$ (n_x, n_y, n_z).

The flight attitude of a thin wing is described by angle of attack α and yaw angle β of the body, and the circumferential direction direction position angle Γ and rudder deflection angle δ_1 of the body relative to the wing. This can be expressed by the component of flow velocity $\vec{v}_-$ in the elastic wing coordinate system. They are

$$v_x = \cos\beta\cos\alpha\cos\delta_1 - (\cos\beta\sin\alpha\cos\Gamma - \sin\beta\sin\Gamma)\sin\delta_1$$
$$v_y = \cos\beta\cos\alpha\sin\delta_1 + (\cos\beta\sin\alpha\cos\Gamma - \sin\beta\sin\Gamma)\cos\delta_1$$
$$v_z = \cos\beta\sin\alpha\sin\Gamma + \sin\beta\cos\Gamma$$

In the equations, by definition Γ is in positive rotation in the clockwise direction as viewed from the body tail facing the wing planform. $|\Gamma| \leq 180°$; by definition, δ_1 is in positive rotation in the clockwise direction as viewed from the wing tip toward its rear fringe.

(3) Calculation of heat flow rate

a. Heat flow rate of laminar flow: Lees' laminar heat flow rate formula [6] can be used in the heat flow calculation with respect to axial symmetry. Using the infinitely long cylinder relation, it is easy to derive the form of Eq. (2). Lees' formula of the heat flow-rate at stationary-point line is

$$q_{s.Lees}=0.5\,g\,Pr^{-\frac{1}{3}}C_p(T_r-T_w)_s G\sqrt{\rho_s\mu_s}\sqrt{\frac{V_\infty}{R_s}}$$

In the equation, T_{rs} is the recovery temperature on the stationary-point line; Pr is the Planck number. However,

$$G=\left[\frac{1}{V_\infty}\sqrt{\frac{2(p_s-p_\infty)}{\rho_s}}\right]^{\frac{1}{2}}$$

Lees' relative heat flow-rate formula is

$$\left(\frac{q}{q_s}\right)_{Lees}=\frac{\sqrt{R_s}}{\sqrt{2G}}\frac{(p_s/p_s)(u_s/V_\infty)\sin\theta}{[\int(p_s/p_s)(u_s/V_\infty)\sin\theta\sin\theta_1 ds]^{1/2}}$$ /401

In the equation,

$$\sin\theta_1=\sqrt{1-\frac{n_1^2\sin^2\Lambda}{n_s^2+n_1^2}}$$

Cohen's drift cylinder relation [7] can be viewed as a revision of Lees' formula. Here, this revision is adopted in part. Hence, the heat flow-rate of the laminar flow stationary-point line, and the relative heat flow-rate are, respectively,

$$q_{s.1}=1.114\,q_{s.Lees}$$

$$\left(\frac{q}{q_s}\right)_1=\left(\frac{q}{q_s}\right)_{Lees}\cdot\beta\beta\cdot\frac{\zeta}{\zeta_s}$$

In the equations, ββ and ζ/ζ_δ are, respectively, the pressure gradient and the revised entropy for external flow.

b. Heat flow-rate of turbulent flow: The Vaglio-Laurin formula [8] can be used for a random three-dimensional object. Using the working formula of an infinitely long cylinder and dimensionless heat flow-rate of Lees' stationary-point line, the relative heat flow-rate of turbulent flow is obtained as

$$\left(\frac{q}{q_s}\right)_i = C_{ti} \frac{(\rho_e/\rho_s)(\mu_e/\mu_s)(u_e/V_\infty)(\sin\theta)^{1/4}}{[\int(\rho_e/\rho_s)(\mu_e/\mu_s)(u_e/V_\infty)(\sin\theta)^{1/4}\sin\theta_i ds]^{1/5}}$$

In the equation,

$$C_{ti} = \frac{0.058\sqrt{R_s}}{G}\left(\frac{\rho_s V_\infty}{\mu_s}\right)^{0.3}\left(\frac{\mu_s}{\mu_0}\right)^{0.3}$$

c. Heat flow-rate of a turning region: In reference [11], experimental results are given for M_∞ = 4.04, with angles of attack as high as 20° and with the front fringe sweptback angle thin the range 0° to 72°. In the reference, the equivalent stationary-point pressure is quoted

$$p_0' = p_s\left(1+\frac{\gamma-1}{2}M_{cn}^2\right)^{\frac{\gamma}{\gamma-1}}$$

In the equation, M_{cn} is the local Mach number component perpendicular to the front fringe of the wing; $M_{cn} = M_g \sin\theta$. The local turning Reynolds number perpendicular to the front fringe of the wing is

$$R_{cn} = \frac{\rho_s \bar{u}_c \bar{s}}{\mu_s}$$

In the equation, $\bar{u}_c = u_c \sin\theta$. In this paper, based on experimental results, the equation for R_{cn} is simulated:

$$R_{cn} = [35.7 - 1.529(M_{cn} - 2.3)^2]p_0' + 1.406\times 10^3$$

As pointed out in the reference, these relations are also suitable for other experiments, needing only some adjustment of their coefficients.

When the heat flow is calculated point by point along s, the corresponding $\bar{s}$ should also be calculated at the same time. In addition, a comparison is made with the $\bar{s}$ calculated from R_{ch}. Once $|\bar{s}| \geq \bar{s}$, appears, the beginning point of the turning is determined. Within the turning region, the weighted function is utilized to obtain the heat flow-rate in the turning zone from q_ℓ and q_t values.

(4) Thermodynamic relations and transport coefficient: Based on /402 the relation for an infinitely long cylinder, the Mach number component of the incoming flow perpendicular to the leading fringe of the wing is introduced

$$M_\infty = M_- \cos \Lambda_\infty$$

In the equation, the equivalent sweptback angle is

$$\Lambda_\infty = \sin^{-1}(v_\infty \sin \Lambda + v_\zeta \cos \Lambda)$$

Based on the adiabatic conditions of an ideal gas, the temperature of the inviscid flow on the stationary-point line at the front fringe of the wing is obtained:

$$T_\infty = T_- \left(1 + \frac{\gamma - 1}{2} M_\infty^2\right)$$

Based on Sutherland's formula, μ_s is calculated from T_s:

$$\mu_\infty = 1.489 \times 10^{-7} \frac{T_\infty^{3/2}}{T_\infty + 113}$$

Under the conditions of subsonic leading edge ($M_n < 1$) and supersonic leading edge ($M_n > 1$), the p_s/p_∞ for these two cases are derived, respectively, by the following formula:

$$\frac{p_s}{p_\infty}=\left(1+\frac{\gamma-1}{2}M_s^2\right)^{\gamma/(\gamma-1)} \qquad (M_s<1)$$

$$\frac{p_s}{p_\infty}=\left(\frac{\gamma+1}{2\gamma M_s^2-\gamma+1}\right)^{1/(\gamma-1)}\left(\frac{\gamma+1}{2}M_s^2\right)^{\gamma/(\gamma-1)} \qquad (M_s\geq 1)$$

According to the status equation of ideal gas, ρ_s is calculated from p_s and T_s.

The outer fringe values ρ_c/ρ_s and T_c/T_x are derived from p_c/p_s based on an isoentropic relation; however, μ_c/μ_s and u_c are given from the following equations:

$$\mu_s/\mu_\infty=(T_s/T_\infty)^{0.76}$$

$$u_s=\sqrt{2009.6(T_s-T_\infty)}$$

(5) Calculation method for the inviscid streamline direction: Due to different ways of calculating an inviscid streamline, different equations for sin θ can be derived. Here, two approximate relations are given.

a. Streamline method I: On the assumption that $\bar{u}_c$ near the stationary-point line linearly increases with $(\phi - \phi_s)$, $\bar{u}_c$ is the velocity component at the outer fringe of the boundary layer perpendicular to the leading edge of the wing. The following equation is then derived

$$\sin\theta=G^2\frac{V_\infty}{u_s}\left(\frac{\bar{s}}{R_s}-\varphi_s\right)$$

b. Streamline method II: According to Newton's theory of lift, following collision the flow is still within the plane constituted by $\vec{v}_\infty$ and $\vec{n}$. However, the flow direction is along the direction of the tangent to the object surface, that is
The unit vector $\vec{\ell}$ is designated as $\vec{l}\,(l_x, l_y, l_z)$. Under this

approximate condition, we have

$$\sin\theta=\sqrt{1-(l_x\sin\Lambda+l_z\cos\Lambda)^2}$$

In calculations given in this paper, these two methods are jointly used. For $|\varphi-\varphi_s|<45°$ of the wing leading edge, streamline method I is used to calculate sin θ beginning from $\varphi-\varphi_s=\pm45°$ The equation for the position ϕ_S of the stationary-point line is

$$\varphi_s=-\sin^{-1}\left(\frac{v_y}{\cos\Lambda_e}\right)$$ /403

In calculating heat flow, integrations of the laminar flow and turbulent flow formulas are designated, respectively, as I and I_t. To ensure that the heat flow value is continuous, and it is required again to give the initial values of I_ℓ and I_t at $\varphi-\varphi_s=\pm45°$.

(6) Pressure calculations: Finally, the mixed pressure method is used in pressure calculations. Facing the incoming flow, the dip angle at the object surface is as follows:

$$\delta=-\sin^{-1}(v_x n_x+v_y n_y+v_z n_z)$$

When $\delta\geq0$, at the wing surface the semiempirical formula for a joint truncated cone and truncated wedge given by reference [10] is applied; the pressure coefficient is

$$C_p=\frac{4}{\gamma+1}(M_{z1}^2-1)/M_\infty^2$$

In the equation, $$M_{z1}=K_c M_\infty\sin\delta+\exp\left[-\left(K_c-\frac{1}{2}K_w\right)M_\infty\sin\delta\right]$$

In the equation, K_c and K_w are functions of the ratio γ of specific heats. At the wing leading edge, the revised Newton formula is used. In addition, the weighted method is used to ensure that the pressure (at the stationary-point line) is continuously transmitted to the wing surface pressure.

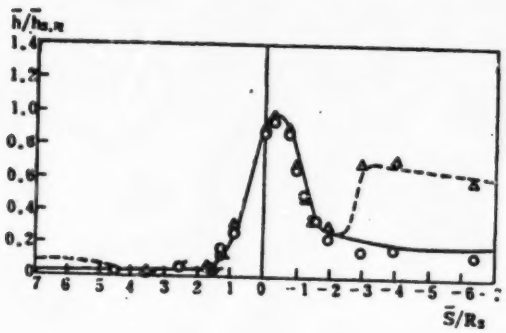

Fig. 3. a

$M_\infty=4.95$; $\Gamma=0°$; $\alpha=10°$; Re_∞/米 $=6.40\times10^6$, 40.16×10^6; b 实验: O, △; c 理论: ——, ---

Key: a. Meter; b. Experimental; c. Theoretical.

When $\delta<0$, the Planck-Mayer expansion wave relation serves in calculating the pressure [10] both at wing leading edge and the wing planform.

III. Comparison Between Theoretical Calculation and Experimental Result

In examining the calculation methods used herein, heat flow-rate calculations are conducted according to the experimental conditions given in reference [4]; a comparison is made with the experimental results.

The experimental model is a plane delta wing with A=60° and R_s=0.0065 meter.

The experimental data are expressed in terms of the heat transfer coefficient with definition as $\bar{h} = q/(T_o - T_w)$

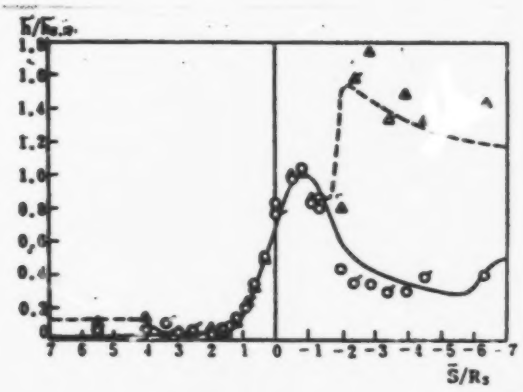

Fig. 4.

a

$M_\infty = 4.95; \Gamma = 45°; \alpha = 20°; Re_\infty/米 = 6.40 \times 10^6$, 40.16×10^6; b实验：○，△；c理论——，---

Key: a. Meter; b. Experimental; c. Theoretical.

In this paper, the heat transfer coefficient $\bar{h}_{s\ \text{theoretical}}$ at the stationary-point line (as calculated theoretically) is dimensionless; the relative heat transfer coefficient $\bar{h}/\bar{h}_{s\ \text{theoretical}}$ is obtained. In Figs. 3 and 4, the variation of the coefficients' $\bar{s}$ are plotted. /404 In the figures, the data marked with a upward slanted hyphen are measured at the points lying along the s direction. For comparison with experimental data, the theoretical results calculated along s are also converted in terms of $\bar{s}$.

In Figs. 3 and 4, the theoretical calculation is consistent with the experimental results; this is satisfactory. This shows that the method has adaptability.

In experiments, the boundary layer is naturally turning. As used in calculations, a coefficient of 0.5 is multiplied by R_{cn}.

Thus, the calculated turning position is calculated as being consistent with the experimental results. This explains that this criterion takes the effect of the sweptback angle and the angle of attack into account.

To explain the effect on calculating heat flow due to the scale factor h_2, Table 1 lists the ratios of θ and $\bar{h}$ calculated separately with two kinds of streamline methods as used herein. In the vicinity ($\bar{s}/R_s$=-0.35) of the stationary-point line, θ calculated with the streamline method I is relatively accurate. On the wing plane, when calculated with the streamline method II, θ=30°. It can be observed that there is a small difference between the calculated heat flow-rates with the two kinds of streamline methods on the wing planform; however, in the vicinity of the stationary-point line of the leading edge, the effect of θ on heat flow is relatively obvious.

Table 1. Effect (α=10° and Γ=0°) on Heat Transfer Coefficient Due to Different Streamline Methods and Different Pressure Methods

$\bar{s}/R_s$		-0.35	-0.98	-1.57	-2.01	-6.00
θ	a 流线法 II/I	1.506	1.390	1.100	1.100	1.100
$\bar{h}$	a 流线法 II/I	1.224	1.136	0.9522	0.9764	1.046
p	b 混合/牛顿	1.000	1.078	1.470	1.476	1.476
$\bar{h}$	b 混合/牛顿	1.000	1.063	1.385	1.355	1.329

Key:

a. Streamline method II/I; b. Mixed/Newton.

IV. Conclusion

(1) This paper utilizes the small cross flow hypothesis and the working formula for an infinitely long cylinder, and a method for calculating the heat flow-rate is obtained. This is adaptable

to blunt-nosed thin wings in supersonic conditions. Satisfactory results are obtained for different angles of attack and yaw angles. In this method, it is not required to calculate numerous inviscid streamlines; thus, the method is relatively simple.

(2) Moreover, in the vicinity of the stationary-point line, different streamline methods have little effect on heat flow-rate calculation. In this paper, two kinds of streamline methods are jointly applied, in obtaining relatively satisfactory results.

(3) The effect of different pressure methods on calculating the heat flow-rate is relatively obvious; so it is important to use the relatively accurate (pressure) calculation method.

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